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X-15 AIRPLANE STABILITY AUGMENTATION SYSTEM

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SUMMARY

This paper describes the basic damper system currently installed in the X-15 airplane, discusses some of the problems encountered during its development and flight testing, and reviews briefly the system reliability.

INTRODUCTION

The proposed performance goals of the X-15 research vehicle made it obvious in the early stages of its development that stability augmentation would be required. In figure 1 the flight envelope of the X-15 is compared with that of a typical century series aircraft. Dampers were necessary in these military aircraft, and it was clear from the estimated speed and altitude that the X-15 would have similar requirements. It was also believed that any system installed to augment the stability should emphasize simplicity and reliability. For these reasons a simple three-axis damper system was proposed which would not include multiple sensors, complicated automatic gain scheduling, or sophisticated automatic control modes.

This paper describes the basic damper system currently installed in the X-15, discusses some of the problems encountered during its development and flight testing, and reviews briefly the system reliability.

SYMBOLS

$C_{1/2}$ cycles to damp to one-half amplitude

$C_{l_{\delta_a}}$ aileron control effectiveness

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L_{δ_a} roll-control power

q dynamic pressure

δ_a total aileron deflection

δ_v vertical-tail deflection

$\Delta\varphi$ peak-to-peak amplitude of limit cycle in roll

ζ damping ratio

SYSTEM CONFIGURATION

A functional diagram of the stability augmentation system (SAS) built by Westinghouse is shown in figure 2. The essential components of the pitch, roll, and yaw channels of the system are indicated as gyros, cockpit gain selectors, electronics, and servos. The outputs of the servos go to their respective control surfaces. Unique features of the system are cockpit gain selection and the inner connection required for operation of the left-hand and right-hand horizontal stabilizers, which provide both pitch- and roll-damper input. Also shown is a yaw rate input to the roll axis. This interconnection is necessary for stability at high angles of attack, primarily because of the high roll input of the lower rudder. The gain-selector settings of 8, 6, 8, indicated for pitch, roll, and yaw, respectively, are the normal settings.

Positive control of failures is emphasized by providing complete fail safety. Figure 3 shows a schematic diagram of the yaw-axis monitoring arrangement which is typical of the pitch and roll channels also. A complete duplication from gyro pick-off to servo input is provided in a working channel and a monitor channel. This arrangement allows a continuous comparison of system performance in the two channels. Automatic shutoff of the affected channel with rapid servo centering is accomplished when a 10-percent variation exists between the working and monitoring channels. Because of the high dynamic performance of the servo cylinder, it was not necessary to duplicate its dynamics in the monitor channel. A simple, constant gain, auxiliary SAS is being fabricated for the X-15, which will serve as a backup in the event of a failure of the primary stability augmentation system.

RELIABILITY

Information concerning the failures experienced thus far with the stability augmentation system is given in figure 4. The number of accumulated failures is plotted against the total number of hours during which the systems have been functioning, both on the ground and in the air. Also, shown in the lower scale is the total number of flights including aborted flights. This scale is necessarily nonlinear because, for example, more ground hours per flight were put on the systems in preparation for the earlier flights than for the later flights. The top curve includes all failures accrued during both ground and flight operation and should be used primarily as logistics information. Note that the failure rate, given by the slope, has greatly diminished. The breakdown given by the bar graph on the right shows the module failures to be the largest single source of system failures. Next is ship's wiring, with miscellaneous other failures accounting for the remainder of the total.

The lower curve represents only the failures which have occurred in flight. "In flight" is defined as the time from take-off of the B-52 to landing of the X-15 or X-15/B-52 combination in case of an abort. A breakdown of the in-flight failures is shown also on a bar graph at the right of the figure. These failures resulted from the malfunction of one electronic module, three instances of broken ship's wiring, and three malfunctioning gain switches. Of these seven, six were traceable to human error and were damaged on the ground but did not result in failures until airborne. Considering only the electronic module failure in 78 flights, and no in-flight failures in the last 13 flights, it can be said with confidence that SAS has proven to be reliable.

LIMIT CYCLES

In the first studies using the X-15 flight simulator, unwanted limit cycles or continuous oscillations, sustained by SAS, were observed. The limit cycles were caused by hysteresis and rate limiting which produce considerable phase lag. The phenomenon was later observed in flight, though at first it was not noticed by the pilots.

An illustration of the magnitude of the limit cycle is shown in figure 5. The peak-to-peak amplitude of the limit cycle is plotted against the roll-control power which is proportional to both dynamic pressure and aileron control effectiveness. The roll damper gain is 0.3 deg/deg/sec or a setting of 6. The circular symbols denote the early flight data. Shown as squares are limit cycles measured on the ground by using an analog computer to close the aerodynamic loop around the X-15 airplane. The solid line gives the calculated limit-cycle characteristics. These calculations were made by using a mathematical model of the nonlinear actuator which included hysteresis, dead band, and rate limiting. Note the extreme increase in the limit-cycle amplitude predicted at large values of control power.

A flight was made to verify these limit-cycle characteristics at large values of this roll-control parameter. Figure 6 shows a time history of roll rate and aileron deflection during the severe roll limit cycle. The frequency of this limit cycle was about 3.2 cps, and the amplitude was about 1° total change in bank angle. This was considered by the pilot to be quite objectionable, partly because of the motion of the control stick caused by surface rate limiting. The amplitude of this limit cycle was not constant, but changed because of control input and a tendency to beat. Figure 7 shows again a comparison of the limit-cycle characteristics obtained in flight and calculated characteristics, with data of the special flight added. Although the critical value of L_{δ_a} appears to be somewhat higher than the calculated value, a drastic increase in bank-angle amplitude would result if the control power were allowed to increase much more.

As a means of reducing the limit cycles to an acceptable amplitude, the SAS electronic filter was modified, which resulted in the limit-cycle characteristics shown in figure 8. The reduced lag of the modified filter greatly reduced the amplitudes of the limit cycles so that the problem was essentially eliminated. The most extreme values of control power did not give objectionable limit cycles.

Although this discussion of the limit cycles has been concerned only with roll, the limit cycles also exist at some flight conditions in pitch and yaw, but to a lesser degree. The limit cycles in pitch and yaw occur at frequencies closely related to the natural frequency of the airplane and do not have the critical nature of the roll limit cycles.

VIBRATION

Although the modified filter greatly lessened the severity of the problem with limit cycles of 1 to 3 cps in roll, a new problem arose. It became apparent during tests on the ground that it was possible to excite and sustain a system-airplane vibration at 13 cps with the modified filter. A breadboard of the modified filter was flown at high damper gains, but the pilot failed to excite the vibration. During the rollout after touchdown, however, a severe vibration was encountered and the SAS had to be turned off. This experience led to the belief that the vibration would occur only on the ground. To prevent recurrence on the ground, a switch which automatically lowered the gain to a safe level when the landing gear was extended was incorporated in the airplane. Five flights later the sense of security engendered was shaken, literally. Figure 9 shows a portion of a time history during reentry from a 170,000-foot-altitude mission. It is obvious that a 13-cps vibration is present in all traces - left and right SAS links, left and right surface deflections, and roll rate. The pilot reported the vibration to be the most severe that he had ever encountered. The shaking was triggered by pilot inputs at low dynamic pressure (130 lb/sq ft) and continued until the SAS gain was reduced slightly and dynamic pressure had climbed to 1,000 lb/sq ft. Fortunately, the amplitude of the shaking was limited by rate limiting of the control-surface actuators. The problem was analyzed to find an explanation for this behavior.

Figure 10 illustrates the mechanics of the phenomenon. The lightly damped horizontal-stabilizer surfaces, represented by the flexible beams with masses, were excited at their natural frequency (13 cps) by the pilot inputs to the control system. The inertial reaction of the fuselage to this vibration was picked up by the gyro, so that the SAS was able to sustain the vibration with inputs to the control surfaces.

Because of the closed-loop nature of the problem, restrictions in the allowable gain exist at the structural frequencies, as shown in figure 11. Presented is system gain as a function of frequency for three filters, all at a SAS gain setting of 6. If the curves intersect these boundaries which represent restrictions in gain at the structural frequencies of the horizontal tail at 13 cps and 30 cps, a sufficient condition exists for a sustained oscillation. The modified filter used during the previously discussed altitude flight intersects the first boundary; a vibration, therefore, would be expected at 13 cps. The original filter now in use is shown to be free of the 13-cps vibration, but produces unacceptable limit-cycle characteristics at critical flight conditions. One way to avoid both problems is to use a notch filter. This filter was designed to give a minimum phase lag at limit-cycle frequencies and a maximum of filtering at the surface resonant frequencies.

An additional modification which would alleviate the problem is a pressure-feedback valve for the surface actuator. This valve would, in effect, augment the structural damping of the horizontal surfaces. Referring again to figure 11, the use of the pressure-feedback valves would lift the restrictions in gain to values outside the range of gain for SAS. Pressure-feedback valves would allow further improvement of the limit-cycle characteristics because of the reduced phase lag associated with removing the notch filter. Both the notch filter and pressure-feedback valve are currently being developed for use in the X-15.

SAS EFFECTIVENESS

The improved handling qualities due to the SAS have been a significant contribution to the success of the X-15 program. Figure 12 shows, as a function of angle of attack and velocity, a significant predicted area of uncontrollability in the lateral-directional modes of the X-15 without SAS. This figure also shows that the SAS should enable control at all except very high angles of attack. References 1 and 2 discuss in detail the control problem involved and suggest methods of analysis.

Figure 13 illustrates the adequacy of the damping of the unaugmented and augmented X-15 airplane in pitch. The coordinates are velocity and altitude, and the solid lines indicate the boundaries of the flight envelope of the X-15 airplane. In this lower region, the pitch damping of the unaugmented airplane is acceptable; less than one cycle is required to damp to one-half amplitude ($\zeta \approx 0.1$). The augmented airplane has as much damping over the entire aerodynamic region. In the ballistic region, aerodynamic control becomes ineffective and the use of reaction controls is required. To provide damping in this region, a reaction augmentation system has been designed and is to be installed in the airplane. This system uses the reaction-control rockets to provide damping about all three axes. Although the airplane can be and has been flown by the pilot without augmentation in the ballistic region, the reaction augmentation system is expected to improve greatly the control characteristics of the airplane. More important, it will provide a good backup damping system for SAS during the setup for the reentry portion of high-altitude flights.

CONCLUSIONS

The design objectives of a simple, reliable stability augmentation system have been achieved. The reliability of the electronic components in flight has been good and approaches the design objective. Limit-cycle problems predicted on the simulator have been verified in flight. A vibration problem not anticipated was encountered in flight with a modified shaping and was traceable to structural SAS interaction. Two acceptable means of eliminating these problems have been developed for incorporation into the X-15 airplane.

In short, the overall experience with the stability augmentation system has been favorable, and the improved vehicle characteristics available with the system have enabled the pilots to investigate with confidence many areas which would have been uncontrollable without stability augmentation.

Flight Research Center
National Aeronautics and Space Administration
Edwards, Calif., November 20, 1961.

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2. Taylor, Lawrence W., Jr.: Analysis of a Pilot-Airplane Lateral Instability Experienced With the X-15 Airplane. NASA TN D-1059, 1961.

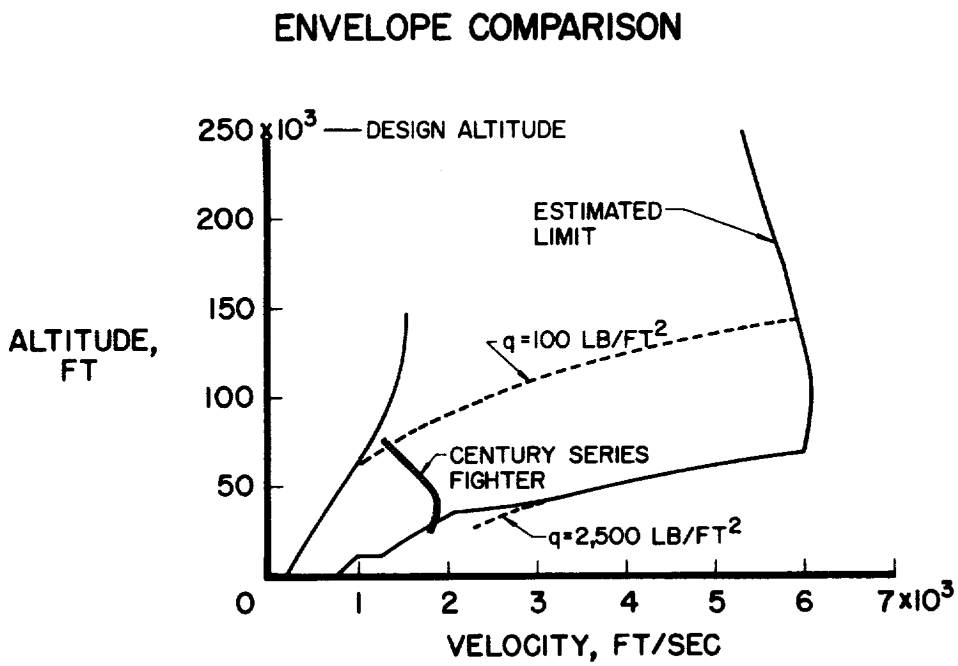

Figure 1

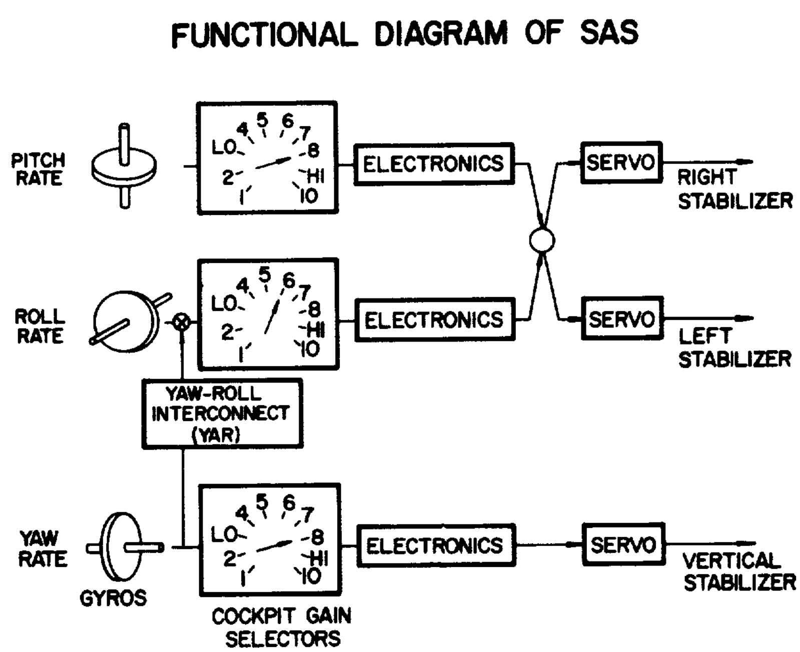

Figure 2

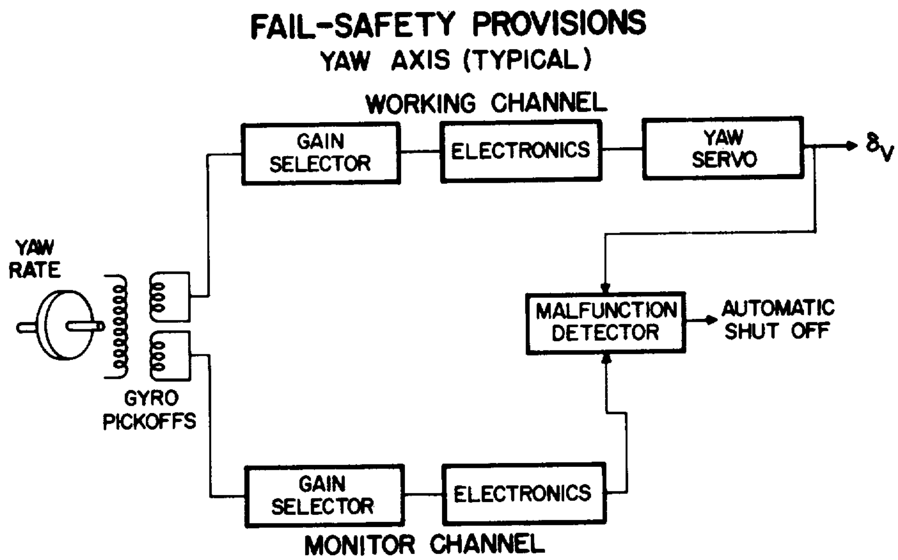

Figure 3

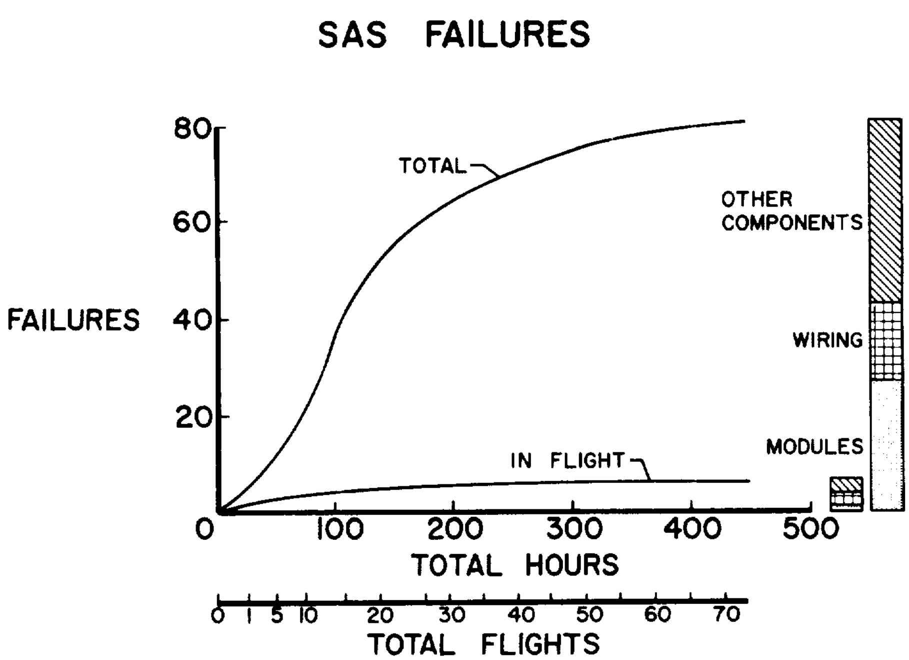

Figure 4

EARLY ROLL LIMIT-CYCLE DATA

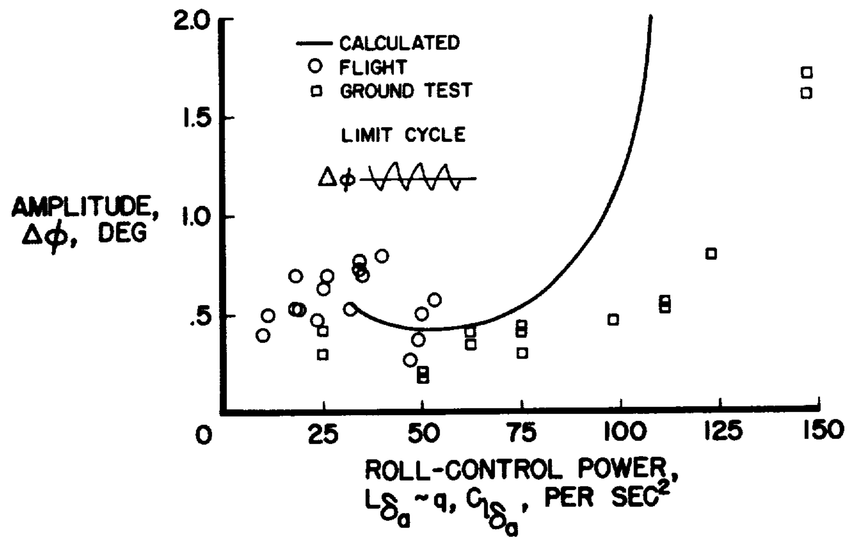

Figure 5

TIME HISTORY OF SEVERE ROLL LIMIT CYCLE

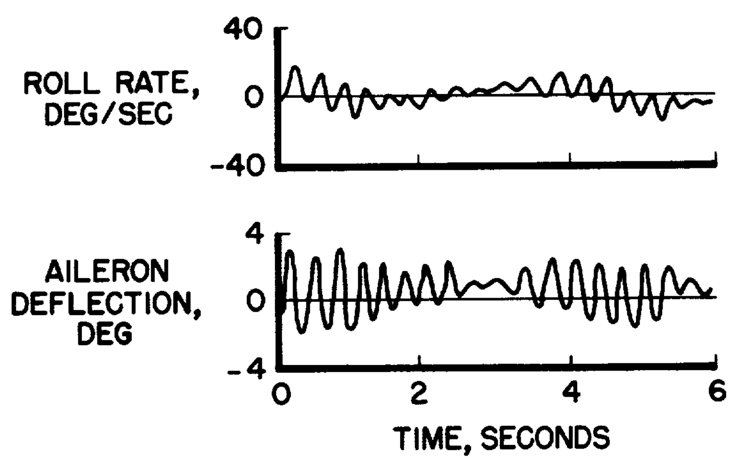

Figure 6

COMPARISON OF CALCULATED AND FLIGHT ROLL LIMIT CYCLES

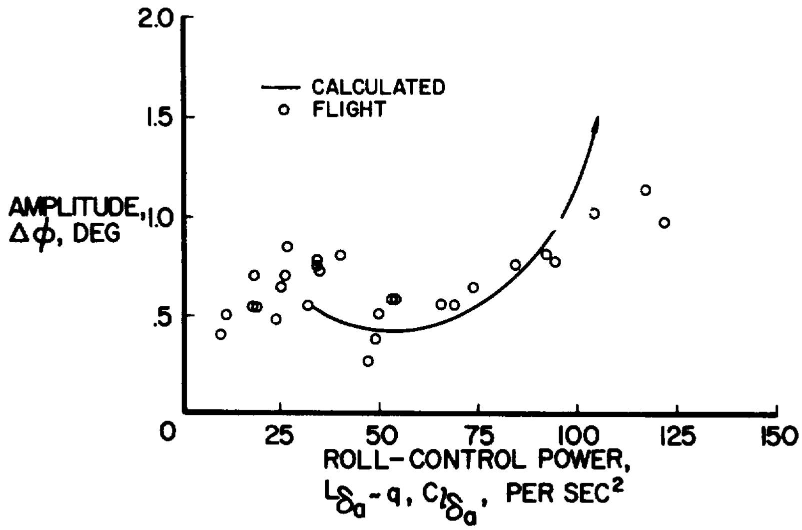

Figure 7

EFFECT OF MODIFIED FILTER ON ROLL LIMIT CYCLE

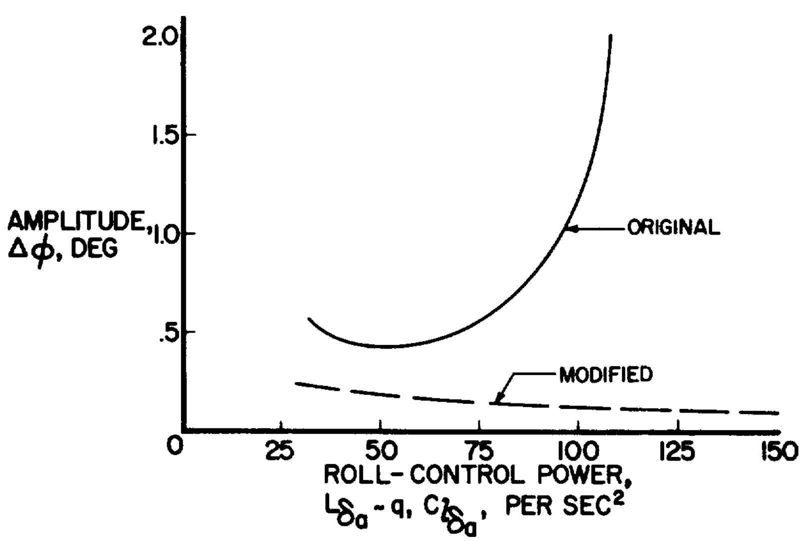

Figure 8

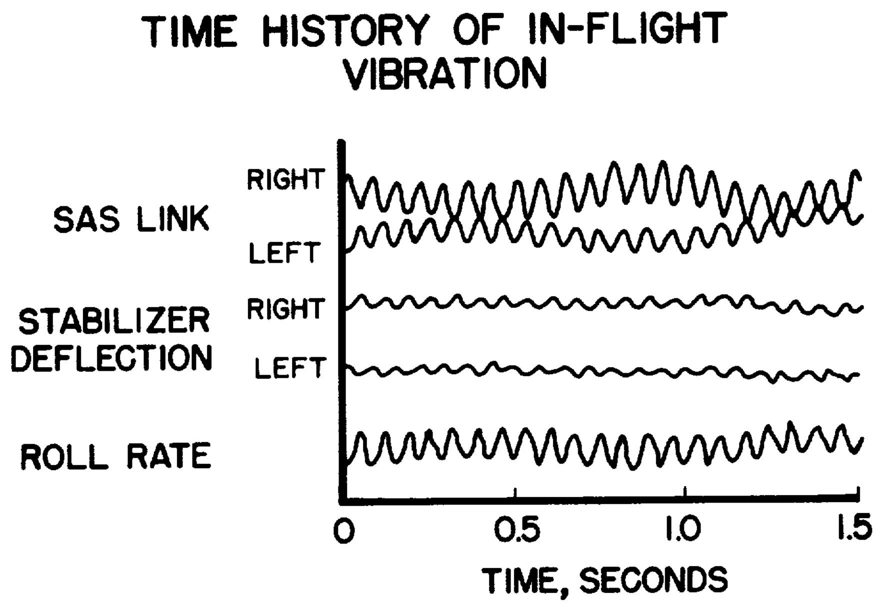

Figure 9

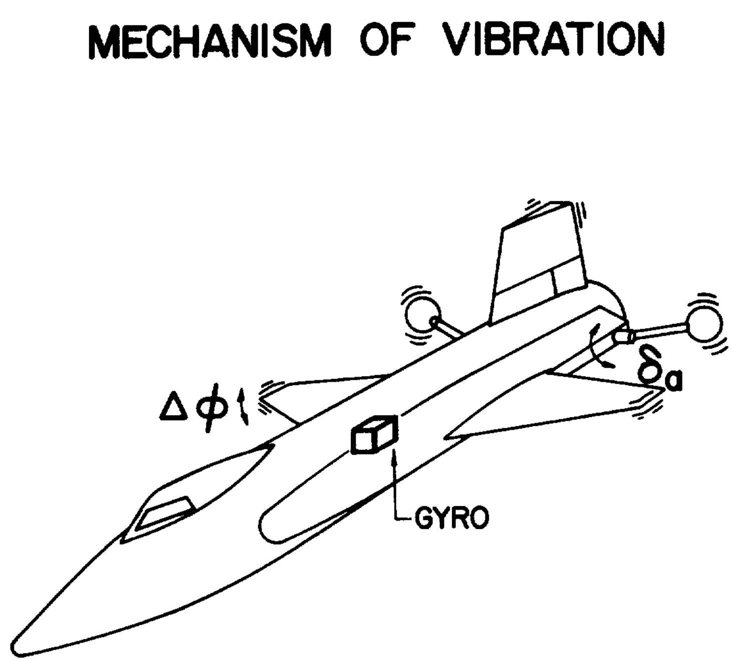

Figure 10

EFFECT OF FILTERS ON HIGH-FREQUENCY STABILITY

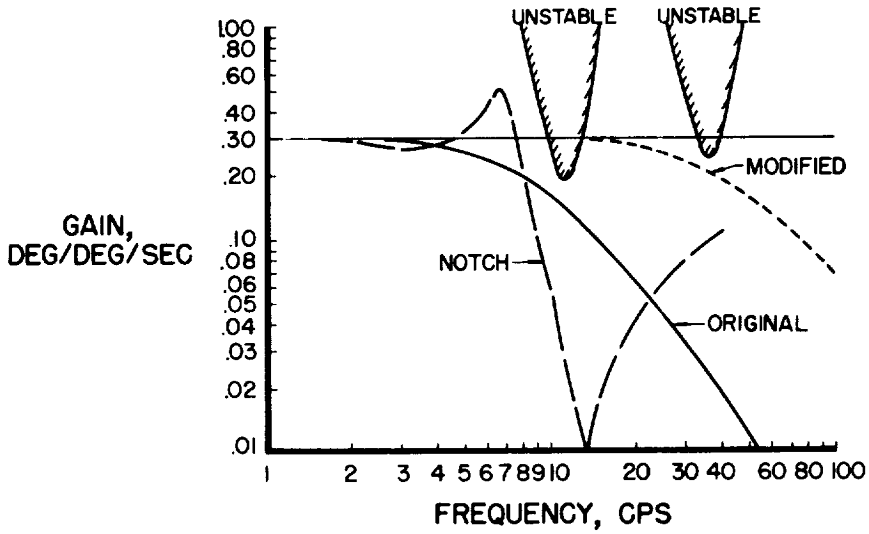

Figure 11

SUMMARY OF LATERAL-DIRECTIONAL STABILITY AND CONTROL

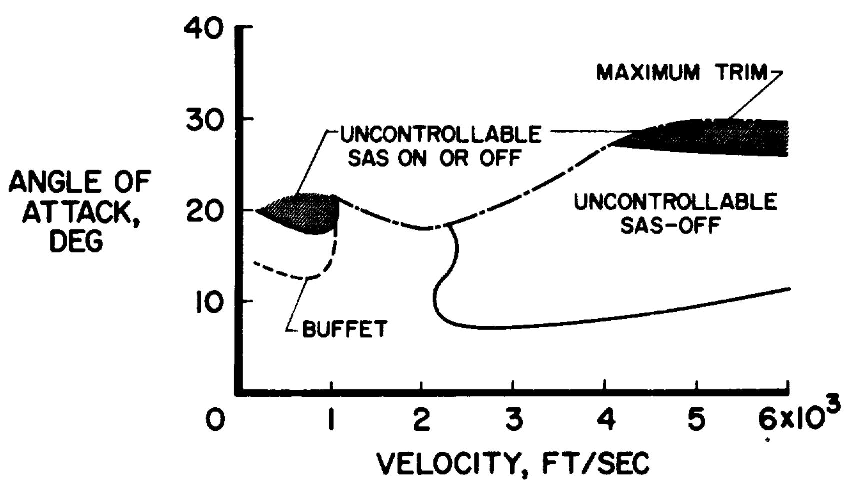

Figure 12

EFFECT OF SAS ON LONGITUDINAL DAMPING

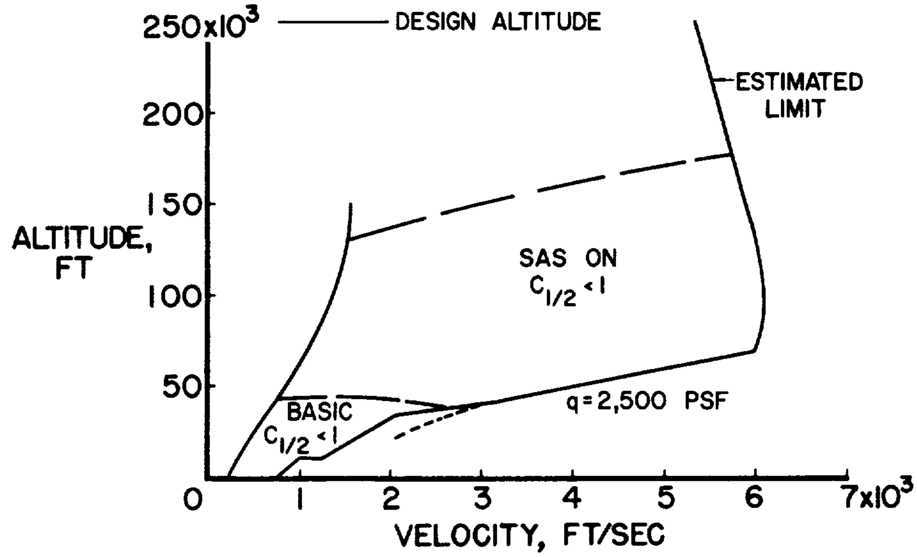

Figure 13